

Chemical Propulsion for Outer Planet Exploration

Arturo R. Casillas,¹ and Jonathan R. Reh²

Jet Propulsion Laboratory, California Institute of Technology, Pasadena, CA, 91109, USA

and

James Gervasi,³ and Shae Williams⁴

Moog Inc. - In Space Propulsion, Niagara Falls, NY, 14304

Several investigators over the last few decades have described the anticipated advantages of implementing chemical propulsion subsystems that operate at lower freezing points, compared to traditional earth storable propellants. For certain missions and for certain spacecraft architectures, the ability to maintain spacecraft at colder Allowable Flight Temperature (AFT) enables drastic reductions in spacecraft mass and mechanical complexity. JPL has assessed the impact of implementing the MMH/MON25 propellant combination (minimum AFT -40°C), as compared to conventional MMH/MON3 (minimum AFT 0°C) or N2H4/MON3 (minimum AFT 10°C). For orbiters to outer planets or ocean worlds and for long-duration, lower-cost solar-powered spacecraft the MMH/MON25 combination can yield several hundred kilograms in spacecraft mass savings. The associated cost savings are estimated to more than offset the requisite propulsion development costs. This paper also presents the results of an exploratory engine test program intended to gain insight into the feasibility of adapting an existing MMH/MON3 flight-qualified engine for operation with MMH/MON25. If successful, this approach has the potential to expedite the path to mission infusion. The engine selected for this purpose is Moog's DST-series engine. Factors that determined this selection include the flight-demonstrated long life and robust performance in terms of long-burn and pulse-mode operation. Hot-fire data is presented with MMH/MON25 propellants at temperatures of 20°C and -40°C . It is found that boiling in the oxidizer passages at 20°C produces stable but lower performance than with MON3. At -40°C the thermal stability and combustion performance are improved but not to the level achieved with MON3 (i.e., 290-sec Isp vs. 300-sec). Possible causes for the observed behavior are discussed.

I. Nomenclature

AFT	=	Allowable Flight Temperature
Cold Prop	=	Propulsion Subsystem based on -40°C minimum AFT (i.e., MMH/MON25)
CHF	=	Critical Heat Flux
ΔV	=	Spacecraft velocity change
EC	=	Europa Clipper
ELF	=	Enceladus Life Finder
HP	=	High Pressure
Isp	=	Specific Impulse
JPL	=	Jet Propulsion Laboratory
LP	=	Low Pressure
MAV	=	Mars Ascent Vehicle
MMH	=	Monomethylhydrazine fuel
MON3	=	Nitrogen tetroxide with 3% Mixed Oxides of Nitrogen

¹ Group Supervisor, Chemical Propulsion and Fluid Flight Systems.

² Mechanical Engineer, Chemical Propulsion and Fluid Flight Systems.

³ Product Line Engineering Manager, Moog Inc. Space and Defense Group, and AIAA Member Grade (if any) for first author.

⁴ Mechanical Engineer, Moog Inc. Space and Defense Group, and AIAA Member Grade (if any) for second author.

MON25 = Nitrogen tetroxide with 25% Mixed Oxides of Nitrogen
N2H4 = Hydrazine fuel
NO = Nitric Oxide
PMD = Propellant Management Device
Rupe No. = Injector Oxidizer-to-Fuel ratio of the (density)-(velocity squared)-(orifice diameter) product
RT = Room Temperature

II. Introduction

As reflected in NASA's most recent Strategic Plan [1], ocean worlds orbiting Jupiter (e.g., Europa, Ganymede, Callisto) and Saturn (e.g., Enceladus, Titan, Dione) are of tremendous interest to scientists for their potential to sustain life. While the Juno and Cassini missions demonstrated the technical feasibility to explore these solar system locations using either solar or radioisotope power systems, the Europa Clipper (EC) project determined [2], that the solar-powered option would be the most cost-effective for its needs, albeit with a mass and complexity penalty. The extreme dearth of radiant energy at these planets greatly exacerbates the solar power supply problem, of course. The sheer size and structural complexity of the necessary solar arrays, especially in the case of Saturn missions, presents significant engineering challenges that require sizable budget allocations to resolve. Further exacerbating this challenge is the prospect that many future missions could likely have increased instrument power needs (EC having roughly twice the solar array area of the 12-year earlier Juno, for example). One way to significantly reduce this engineering and cost challenge is to cut the power requirement without compromising the mission design or instrument suite. This can be done by utilizing lower-temperature capable propellants. For EC, for instance, approximately two-thirds of the roughly 450-Watt power requirement at Jupiter is driven by the need to maintain the spacecraft above the propellant freezing temperature³. Since the heat loss to space is a fourth-power relationship, it can be cut in half by reducing the minimum AFT from roughly 10°C, as in Juno, to -40°C, as proposed herein. Thus, this would reduce the total power requirement by roughly 1/3. The propellants proposed are MMH and MON25. These are the same propellants used in the EC spacecraft except that the oxidizer contains 25% NO instead of 3%. This increase in NO suppresses the freezing point to roughly match that of the MMH fuel and enables the 50°C-lower AFT, without compromising engine performance. A September 2017 Team-X study at JPL quantified the effects of using MMH/MON25 versus N2H4/MON3 propellants in the ELF mission to Saturn (Fig. 1). This spacecraft-level trade confirmed the anticipated 1/3-reduction in solar array power (230 vs 325 W) and in solar array area (73 vs 109 m²), and a 215-kg reduction in spacecraft mass, as well as a net reduction in project phases A through D costs, including the cost of MMH/MON25 technology development.

It is emphasized, however, that reducing the spacecraft minimum AFT does not necessarily yield these benefits for other solar-powered missions. Team-X also analyzed a lander mission in which orbit insertion was accomplished with solid propulsion and it did not yield nearly the same magnitude benefits. Thus, MMH/MON25 would not necessarily be the best choice for landers into bodies with an atmosphere or those that accomplish the orbit insertion maneuver without chemical propulsion. The missions that benefit from this strategy are those with relatively large propellant tanks (i.e., large ΔV accomplished with chemical propulsion), as in all the missions mentioned thus far in this paper.

The remainder of this paper will refer to implementation of the MMH/MON25 propellant architecture at -40°C minimum AFT (approximately -50°C min qualification) as "Cold Prop" in order to distinguish it from earth-storable MMH/MON3 or N2H4/MON3 on the one hand, and cryogenic propulsion on the other.

³ Internal JPL Presentation. Art Casillas. "Dual-Mode vs. Bipropellant Propulsion Subsystem Trade". 7/13/2013.

⁴ Internal JPL Report. Team-X. "1942 Prop Swap for ELF. 2017-09 FINAL Report". September 12 and 14, 2017.

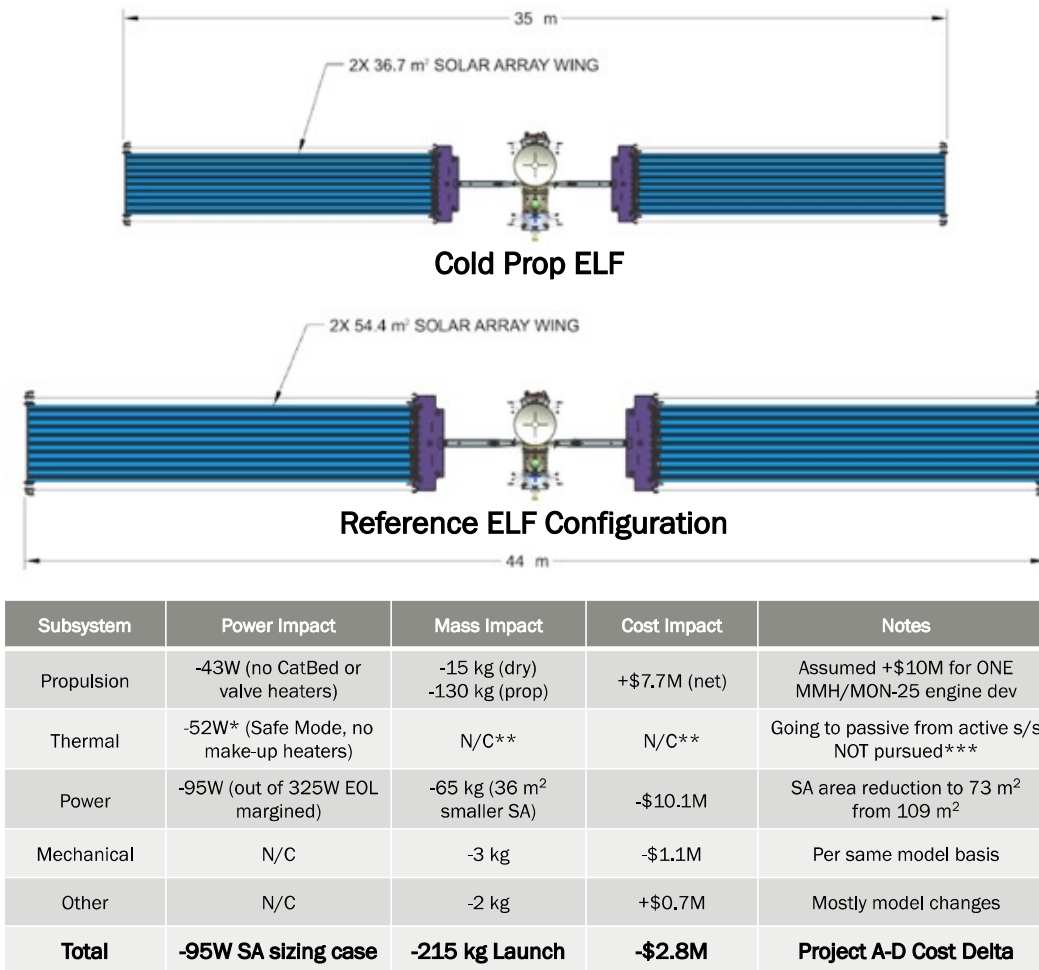


Fig. 1 Cold Prop vs. Dual-Mode Propellants Trade Results for Enceladus Life Finder (ELF) Spacecraft

III. Cold Prop Subsystem Architecture

A survey of typical components used in bipropellant propulsion subsystems shows that the single most important component development necessary to implement Cold Prop is the engine(s) used. Usually a bipropellant subsystem is comprised of HP and LP zones, with a pressure regulator component in-between. Pressurant tanks and HP components are already qualified to -50°C and below due to the common pressurant blowdown cooling effect encountered in most bipropellant systems. The LP propellant tanks, latch valves, service valves, and filters are already typically qualified to the -30° to -50°C range. In the case of those LP components that fall short, HP equivalent ones can be used in their place to benefit from the lower temperature capability afforded by Kel-F (vs. Teflon) seals – at a minor mass penalty. The addition of NO to the oxidizer presents no new manufacturing or material compatibility issues as compared to those of MON3. The colder temperature does affect the design of the surface-tension PMD within each propellant tank. However, as surface tension increases with cooler temperature it is expected that a PMD designed for operation at RT will have superior performance at colder temperature. The PMD design process, in any case, is well-known and once the requisite density, viscosity, surface tension and liquid contact angle are measured for the design material and surface conditions over the -50°C to 50°C qualification range the propellant management function presents no new technology challenges. The situation is similar for fracture mechanics considerations in the propellant tanks. The threshold and environment-assisted stress intensity value(s) need to be measured for the tank material per ASTM E1681 to ensure compliance with safe-life requirements. But the process is low-risk engineering-to-go task rather than

new technology to be developed. In the case of the bipropellant engine(s), however, the injector mixing and atomization process is highly dependent on the propellants' vapor pressure, transport properties, and actual injection conditions. The resulting combustion heating pattern, moreover, must be managed by a combination of film cooling and radiation cooling that also depend on the injection performance, not to mention the thermal transients that must be survived over a wide range of conditions. Characterizing these sensitivities requires extensive hot-fire testing to understand a particular design's performance trends. There is no low-risk standard design or test approach to ensure success. Therefore, the development of the engine(s) necessary for a particular mission represents the top technical risk for implementation of the Cold Prop architecture. It is for this reason that the remainder of this paper focuses on the main engine aspects of Cold Prop.

IV. Previous MMH/MON25 Engine Hot-Fire Test Overview

The concept of employing MMH/MON25 propellants to reduce the power requirements of a spacecraft is not a new idea, but the technology has not been advanced beyond the TRL 4 level (i.e., component validation in laboratory environment) thus far. Two factors warrant a fresh look at this technology, however. First, recent advances in solar power subsystems have shown the feasibility of flying much less expensive non-nuclear powered missions beyond the inner solar system (e.g., Juno). Second, NASA's stated goal of exploring Ocean Worlds and Ice Giants [1] calls for a reassessment of ways to design spacecraft to operate with the diminished solar energy at these destinations.

In the late 1990s Mars Ascent Vehicle (MAV) trade studies favored Cold Prop as a near-optimum balance between high performance and development risk when compared to other Earth-storable and cryogenic propellant options [3, 4]. These assessments led to successful hot-fire testing of the then Kaiser Marquardt (now Aerojet Rocketdyne) R-6C 5 lbf thruster, designed for MMH/MON3 operation at 20°C [5]. Initial concerns associated with delayed hypergolic ignition due to suppressed vapor pressure at the low temperature, potentially leading to hard starts, were investigated. Seven 2-second hot-fire tests at 20°C and -40°C were run. "No significant ignition delay was observed at -40°C compared to ambient propellant temperature". Further, these initial tests evidenced similar mixing and combustion behavior of this propellant combination as compared to more conventional propellants. Forty-four hot-fire tests at -40°C ranging in duration from 10 to 420 seconds were run to map out the performance of the single unlike doublet injector. "The optimum performance of the engine was achieved at a Rupe number of unity". This conclusion was based on combustion chamber pressure readings as thrust was not directly measured. This test program also demonstrated the test facility process for the safe production of MON25 from MON3 by simply adding NO gas while managing the homogeneity and temperature of the resulting oxidizer.

Shortly after the R-6C engine test, the Glenn Research Center (GRC) selected MMH/MON25 for the Mars Flyer Program for much the same reasons as JPL did for MAV [6]. The Atlantic Research Corporation (ARC), now Moog In-Space Propulsion, tested the MMH/MON3 2-lbf Low Thrust Thruster (LTT) with MMH/MON25 for the Mars Flyer Rocket Propulsion Risk Assessment Program [7, 8]. As with the R-6C testing, no modifications were made to the engine to accommodate the new oxidizer. This LTT test showed nearly the same Isp as with MMH/MON3 and essentially unchanged Isp performance with propellant temperature, based on thrust measurements. It should be noted that at the time an apparent 10-sec Isp drop from 20°C to -40°C, attributed to mixture ratio increase due to the MMH viscosity increase, was reported. More recent analysis of the data indicates, however, that this apparent trend was biased by a single out-of-family data point, and that the Isp trend was in fact flat with propellant temperature, even with the mixture ratio drift observed. This significant finding suggested that an existing engine qualified and flight-proven with MMH/MON3 might be adapted for MMH/MON25 service with relatively minor modifications.

Around 2010 a MSFC/APL team selected the MMH/MON25 combination for the Robotic Lunar Lander Development (RLLD) project, again for the low temperature capability [9, 10]. Unlike the previous efforts, however, this project initially selected engines originally developed for tactical missile applications, rather than for in-space service. As these engines are designed for much higher pressures, much faster response, and much shorter life times than required for deep-space exploration, the applicability of the resulting test data to our present purpose is limited and was not considered further in this survey.

More recently there have been engine development efforts that have shown some promise, albeit at very limited test conditions thus far. These include the ISE-100 engine tested by Aerojet Rocketdyne [11], the Deep Space Engine (DSE) work carried out by Frontier Aerospace [12] and the AGILE Space Propulsion line of Advanced Space Engines (ASE) development work [13]. Unfortunately, these efforts currently lack the specific mission pull and corresponding funding commitment needed to infuse the Cold Prop architecture into a flight mission in the foreseeable future.

⁷ Internal JPL Presentation. Art Casillas. "LTT MON-25 Test Report. Fresh Look at 1999 LTT Data for Potential Application to 2018 DST-13 Test Series". 6/19/2018.

V. DST-Series Engine MMH/MON25 Test Description

Given the estimated spacecraft-level benefits of Cold Prop and the state of engine development just described, JPL decided to embark upon a test program aimed at finding out whether a flight proven MMH/MON3 engine could be qualified for MMH/MON25 service with little to no modifications. If successful, this approach would bypass many development risks – and costs – associated with engine material selection, manufacturing, and structural issues, resulting in the most expedited path to mission infusion. JPL chose Moog’s 5-lbf DST-series engine for this test program. The factors that led to this decision include: (1) The DST-13 is the engine selected for the EC spacecraft, which has a mission design and set of operational requirements similar to the type of Ocean Worlds exploration missions that most benefit from the Cold Prop architecture. (2) The DST-13 engine thrust level, specific impulse, pulse capability, and operating box are sufficient to perform large ΔV and small attitude-control maneuvers without the need for additional engine types, thus reducing cost and complexity. (3) Reliability: there are hundreds of DST-series engines flying with no reported failures or anomalies. (4) Robustness: The unique Pt/Rh thrust chamber material has enabled this engine’s exceptionally long life with virtually no operational “keep out zones”.

The goals of the MMH/MON25 test were to demonstrate (1) Isp of at least 300 seconds, (2) stable thermal behavior in steady and pulse-mode operational modes, and (3) combustion stability over a wide range of temperature and inlet pressure conditions. The nominal thrust level of the test engine was set at 6.2 lbf (28 N), similar to the EC engine. The nominal mixture ratio was set at 1.59, consistent with these propellants’ density ratio assuming a 20°C loading scenario.

To investigate how a DST-series engine would perform with MMH and MON25 at -40°C, Moog assembled a flight-like engine out of existing components (Fig. 2). The injector chosen is from a DST-12 engine which is one model in the DST family that is designed for MMH/MON3 operation with a dual-seat torque motor valve. Other DST engines operate with either two or a single solenoid valve per propellant side. These differences in valve configuration have a minimal impact on pulse performance and do not affect steady-state behavior. The DST-12 engine used in this testing includes a welded chamber and injector assembly that were individually previously tested at Moog’s Niagara Falls facility with MMH and MON3, enabling a direct comparison with the same hardware between MON25 and MON3 operation. Only the trim orifice sizes were different between the configurations, as necessitated by the higher than typical thrust target.

All testing was performed at Moog’s Niagara Falls facility in the high-altitude test cell A-1. This cell is typically used for hypergolic bipropellant testing, and during this test a mechanical pumping system was used, reaching low pressure to simulate >120,000 feet of altitude. Moog upgraded this test cell with the capability to feed very low temperature propellants in steady state testing, demonstrating as low as -46°C on both fuel and oxidizer sides. The capability to prechill hardware was also upgraded and used in this testing. Moog assembled and conducted typical preliminary tests such as water flow and proof pressure checks in a clean room environment. The water flow tests confirmed the basic atomization spray pattern and the injector hydraulic performance was in-family and consistent with the original test results. Environmental testing was not conducted since the goal of this testing was a high-level characterization of engine performance with a new oxidizer. The test cell was instrumented with flow meters, pressure transducers, and other instruments as listed in Table 1. Due to the potential impact on pulse-mode performance, there was no chamber pressure transducer port but the delivered thrust was directly measured. An error analysis was carried out that gave the total error measurement of thrust from all sources for this testing. At the three standard deviation level, the measurement error of thrust was calculated at 0.042 lbf (0.68% of



Fig. 2 DST-series Engine Tested with MMH/MON25

Table 1. Instrumentation List for DST-12 MMH/MON-25 Testing.

Parameter	Symbol	Instrument Type	Range	Accuracy	Units
Fuel Flow	WF 1-2	Micro Motion CMFS010	0.003-0.05	0.10%	lbm/s
Oxidizer Flow	WO 1-2	Micro Motion CMFS010	0.003-0.05	0.10%	lbm/s
Thrust	F-A, F-B	Sensotec 41/9735-01	0-25	0.40% (at full scale)	lbs
Fuel Feed Pressure	FFP 1-2	Taber inst. 226-500A	0-500	0.75%	psia
Ox Feed Pressure	OFP 1-2	Taber inst. 226-500A	0-500	0.75%	psia
Fuel Line Pressure	FLP 1-2	Taber inst. 226-500A	0-500	0.75%	psia
Ox Line Pressure	OLP 1-2	Taber inst. 226-500A	0-500	0.75%	psia
Fuel Inlet Temp	FIT	CR-AL-20	-310 to 1017	.75% (4°F min)	°F
Oxid Inlet Temp	OIT	CR-AL-20	-310 to 1017	.75% (4°F min)	°F
Fuel Tank Pressure	FTP	Taber inst. 226-500A	0-500	0.75%	psia
Ox Tank Pressure	OTP	Taber inst. 226-500A	0-500	0.75%	psia
Nozzle Exit Pressure	NEAP 1-2	MKS 626B Baratron	0-10	0.50%	mbar
Valve Voltage	DVAL-E	Calibrated Voltmeter	0-40	0.75%	V _a
Valve Current	DVAL-I	Calibrated Ammeter	0-1	0.30%	A
Injector Backface Temp	IBT 1-2	CR-AL-50	-310 to 2300	2.10% (4°F min)	°F
Valve Body Temp	VBT 1-2	CR-AL-20	-310 to 1017	.75% (4°F min)	°F
Test Cell Ambient Temp	Tamb	CR-AL-20	-310 to 1017	.75% (4°F min)	°F
Pyrometer	---	Mikron M90	900-3000	0.70%	°C
Thermal Camera (Mikron)	---	Mikron M9103	600-1700	1.00%	°C
Thermal Camera (FLIR)	---	FLIR SC6700	-20 to 2000	1.00%	°C

nominal value), of steady state specific impulse at 2.1 seconds (0.71% of nominal value), and of mixture ratio at 0.0013 (0.08% of nominal).

VI. DST-Series Engine MMH/MON25 Test Results and Discussion

Moog and JPL collaborated to plan a full test matrix with 25 steady state and 12 pulse-mode tests. This test matrix was developed with statistical design methods to be able to fully characterize the interaction of propellant temperatures, feed pressures, and mixture ratios. The design space was from -40°C to +40°C, and the goal was to obtain data between a 1.30 and 1.90 mixture ratio focusing around the 6.20 lbf mark. From the outset of testing, however, unexpected emissions from the test cell exhaust stack were observed, leading to a testing shutdown directed by Moog's environmental and safety personnel before the test matrix could be completed. The exhaust emissions were attributed to incomplete combustion of the MMH/MON25 propellants.

Six hot-fire tests were conducted in total, 3 steady state tests at ambient (20°C) and 3 steady state at low (-40°C) propellant temperature. Table 2 shows a summary of the test conditions run and the outcome of each. The highest Isp measured was 290 seconds with -40°C propellants. There were clear indications of oxidizer boiling in the injector for the 30-second 20°C tests. There were no signs of thermal instability or propellant boiling in the injector for all of the -40°C tests performed. There were no signs of combustion instability or hard starts and redline temperatures were not approached for any of the tests run. Note that since the chamber pressure was not directly measured it was inferred from the injector water flow conductance measurements done prior to the hot-fire tests. This extrapolation is adequate for injector conductance comparison with the 20°C hot-fire data but can only be used for relative trend assessments when comparing with the -40°C data due to the propellant viscosity changes. Fig. 3 plots the Isp data at equivalent time slices. The 20°C tests showed decreasing Isp performance as time went on, and higher Isp with increased fuel injector momentum relative to the oxidizer. In contrast, with -40°C propellants Isp was noticeably higher and much less dependent on mixture ratio and time.

Table 2 Summary of Tests Run and Results

		RAW TEST DATA												REDUCED DATA							EST FROM WATER FLOW DATA											
Test (A1-565...)	Date (9/18)	TIME sec	WF	WO	F	FFP	OFF	FIT	OIT	IBT-1	IBT-2	TI Temp	WT	Global	MR	ISP	Density lb/ft3	Rupe	Ts@Pc+Pdyn - F	Pc	Inj Vel - ft/s	Dyn P - psi	Dyn P/Pc	Koxinj								
			lbm/s	lb/s	psia	270	270			deg F			lbm/s		Injector		fu	ox	No	fu	ox	psia	fu	ox	fu	ox	HF	WF				
59	11	10	0.0081	0.0124	5.70	270	270	66	66	138	136	2,220	0.0205	1.52	2.21	277.3	54.7	87.0	0.90	378	119	145	105	64	65	39	45%	27%	137	139		
60	11	10	0.0082	0.0125	5.79	270	270	66	66	132	133	2,310	0.0207	1.53	2.23	280.2	54.7	87.0	0.91	378	119	144	105	65	65	39	45%	27%	138	139		
		30	0.0083	0.0123	5.65	271	269	66	66	238	210	2,130	0.0207	1.49	2.16	273.7	54.7	87.0	0.85	378	118	141	107	64	67	38	48%	27%	134	139		
		180	0.0083	0.0123	5.54	271	271	67	67	256	227	2,099	0.0206	1.47	2.14	268.6	54.7	86.9	0.84	377	117	139	107	64	68	38	49%	27%	131	139		
61	11	10	0.0078	0.0132	5.49	258	286	68	68	129	129	2,097	0.0210	1.69	2.45	261.3	54.6	86.8	1.10	375	120	142	101	68	60	44	42%	31%	136	139		
		30	0.0080	0.0131	5.39	257	285	69	69	239	212	2,086	0.0210	1.64	2.38	256.2	54.6	86.7	1.04	373	118	137	103	68	62	43	46%	31%	131	139		
		180	0.0079	0.0130	5.33	255	284	69	69	261	231	2,070	0.0209	1.64	2.38	254.7	54.6	86.7	1.04	372	118	136	102	67	61	42	45%	31%	131	139		
62	12	10	0.0077	0.0131	5.92	267	269	-49	-41	32	39	2,491	0.0208	1.70	2.47	284.9	58.4	94.6	1.10	376	121	150	93	62	54	39	36%	26%	144	139		
63	12	10	0.0085	0.0130	6.25	299	280	-46	-41	53	50	2,399	0.0215	1.52	2.21	290.5	58.3	94.6	0.88	385	123	158	103	62	67	39	42%	25%	141	139		
		30	0.0085	0.0130	6.28	299	279	-47	-41	146	117	2,432	0.0216	1.53	2.22	291.3	58.4	94.6	0.89	385	123	159	103	62	66	39	42%	25%	143	139		
64	13	10	0.0090	0.0121	6.12	308	260	-46	-36	79	64	2,321	0.0211	1.35	1.96	290.7	58.3	94.3	0.69	387	120	155	108	58	73	34	47%	22%	142	139		
		30	0.0089	0.0123	6.13	308	260	-46	-37	158	123	2,342	0.0212	1.38	2.00	288.8	58.3	94.4	0.72	387	121	155	108	59	73	35	47%	22%	146	139		

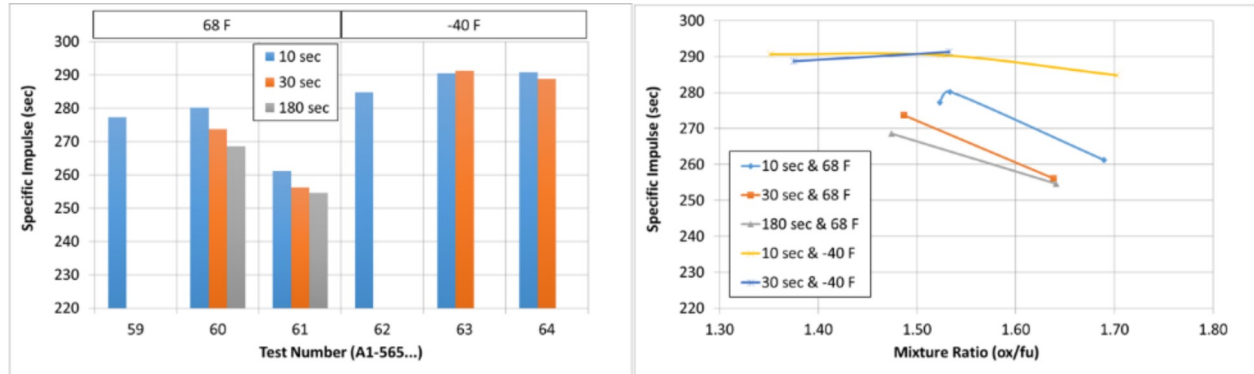


Fig. 3 Isp vs. Time and Mixture Ratio Results Summary

A. DST-series Engine Performance with 20°C MMH/MON25 Propellants

The engine performance at 20°C was characterized by thermally-driven inflection points marked by decreases in combustion performance and thrust chamber temperature and an increase in injector body heat flow. This is illustrated in Fig. 4 which shows the transient data for test number A1-56560. This test had two anomalous inflection points, occurring at 11 and 40 seconds into the 3-minute burn. The decrease in measured thrust (and Isp) at these times is accompanied by (a) a decrease in thrust chamber temperature, (b) an increase in injector body heat flow evidenced by an increase in those temperatures, and (c) an increase in the oxidizer injector pressure drop as evidenced by a decrease in that conductance. These trends are consistent with the unsteady presence of bubbles in the oxidizer injector, caused by thermal transients in the injector plate. At the start of the run the steep rise in chamber temperature suggests a trajectory toward high performance and liquid-on-liquid injection. Note that at this point the oxidizer conductance roughly matches the water-flow measured conductance indicating nominal (i.e., liquid phase) velocity and pressure drop. At 11 seconds the heat wave from the combustion process appears to push the oxidizer injector wall temperatures into the nucleate boiling region, creating two-phase flow, increased oxidizer injection velocity, and loss of injector oxidizer-to-fuel momentum balance resulting in performance drop, decrease in combustion chamber temperature, and rise in injector body heat flow. Note that this condition almost stabilizes at 20 seconds, suggesting a near-equilibrium thermal state was reached between the heat removed by the oxidizer flow at the injector and the heat flow into the injector plate from the combustion process. As the run continues and the engine heats up further, however, the oxidizer injector appears to experience the onset of film boiling (i.e., past the CHF point) with the attendant loss of cooling heat transfer, resulting in irreversible loss of injector momentum balance and poor mixing/combustion performance.

For an excellent empirical demonstration of the impact of the heat-transfer process just described on an unlike doublet injector, the flow patterns and the resulting atomization and combustion performance, the reader is referred to the work carried out by Matsuura, et.al. [14]. That work shows that the Critical Heat Flux (CHF) for MON3 – presumed similar to that of MON25 – occurs when the wall temperature is 40 to 50°C hotter than the oxidizer saturation temperature at that pressure (this point being a weak function of velocity, subcooling, and wall conditions as well). For all six tests run in this series, that saturation pressure was 50°C, or 120°F (Table 2). Thus, the CHF is reached at injector wall temperatures exceeding 90 to 100°C (about 195 to 210°F). This is believed to be the threshold exceeded

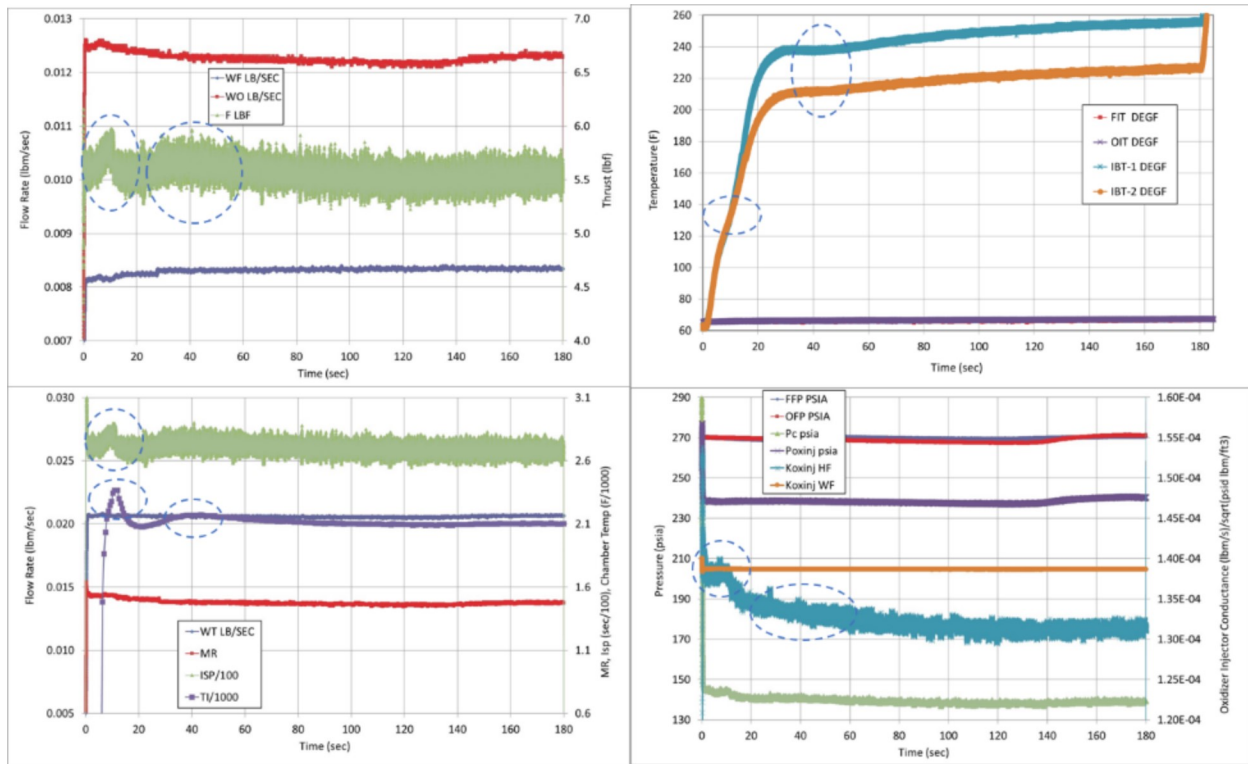


Fig. 4 Test A1-56560 (20°C) Data Showing Two Performance Inflection Points

at 40 seconds into test -60. By way of comparison, using the same criteria the CHF threshold for the injector wall temperature for MON3 is about 30°C higher, or 130°C (about 270°F).

The following 20°C run (-61) was done at higher mixture ratio (1.64 vs. 1.49) in an effort to more closely match the nominal target value. With the increased oxidizer momentum relative to the fuel the same two-point inflection behavior was observed except that it occurred earlier, at 1 and 13 seconds, suggesting the heat flow into the injector plate increased much more than the heat removal by the propellants. The fact that Isp was higher with increased fuel momentum is consistent with the interpretation that the oxidizer injector velocity was higher than intended due to two-phase flow. To appreciate the injector performance sensitivity to two-phase flow, note that since the liquid-to-vapor density ratio of MON25 at the test conditions is of the order of 100, it takes only one percent gas content to drop the net oxidizer density to 50% of the intended liquid value, doubling the injector velocity and resulting in a four-fold increase in momentum. So it doesn't take much boiling to disturb the injector pattern and therefore the combustion performance.

Note that the foregoing is *not* simply the result of cavitation due to MON25's high vapor pressure compared to MON3; it is a coupled heat-transfer issue. At 20°C the team did *not* expect to have performance issues due to the high vapor pressure per se. The vapor pressure of MON25 at 20°C is 60 psia, and with the target thrust of 6.2 lbf a chamber pressure of approximately 150 psia was predicted, for a cavitation margin of 90 psi. In 2007, Moog conducted a delta-qualification test for another customer of the DST-12 design that incorporated steady state test runs with elevated temperature MMH and MON3. That engine saw propellants heated as high as 70°C (about 160°F) and it experienced a downward mixture ratio shift at constant inlet pressures, but very little performance loss (see Fig. 5). With a chamber pressure of about 120 psi roughly matching the MON3 vapor pressure at 70 °C, this earlier testing had a cavitation margin of zero.

As stated before, these low-performance indications were matched by observations of orange-brown emissions from the test cell exhaust during the burns, which caused Moog to cancel the rest of the planned ambient temperature burns and proceed to low temperature tests.

B. DST-series Engine Performance with -40°C MMH/MON25 Propellants

In contrast to the 20°C, the -40°C tests were steady and showed no indications of oxidizer injector boiling. A detailed look at representative run A1-56564 is shown in Fig. 6. There was an inflection in the injector body temperature at about 6 seconds into the burn indicating increased net heat flow into that region, similar to the 20°C tests, but this was accompanied by an increase in the delivered thrust and by a rise in the oxidizer injector conductance, suggesting a decrease in oxidizer injector gas content, rather than an increase. This trend suggests that it took about 6 seconds for any oxidizer vapor initially present to be expelled out the injector. Note that the chamber temperature stabilized after about 5 seconds, albeit for this relatively short 30-second run. While Isp clearly improved over the 20°C runs, it still fell 10-seconds short of the minimum target and the dark gas emissions were still observed coming out of the facility vent, suggesting incomplete combustion.

The sensitivity of Isp to increased fuel momentum (lower mixture ratio) was not nearly as strong as for the 20°C runs, suggesting the Isp shortfall was not caused by injector momentum imbalance alone. Said differently, it was not caused by higher-than-intended oxidizer flow velocity as for the 20°C runs.

There was also an unexplained difference in the combustion roughness among the three -40°C runs – see Fig. 7. Run -63, closer to the nominal mixture ratio, was noticeably rougher. No adverse effects were noted as a result,

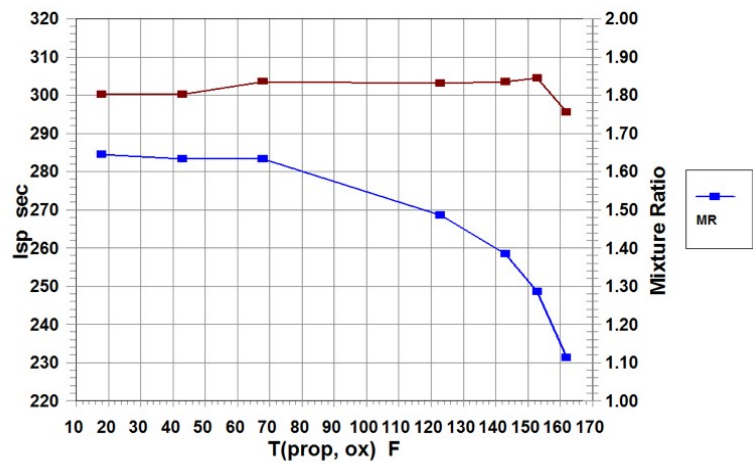


Fig. 5 Previous Testing of a DST-12 Engine with Warm Propellants

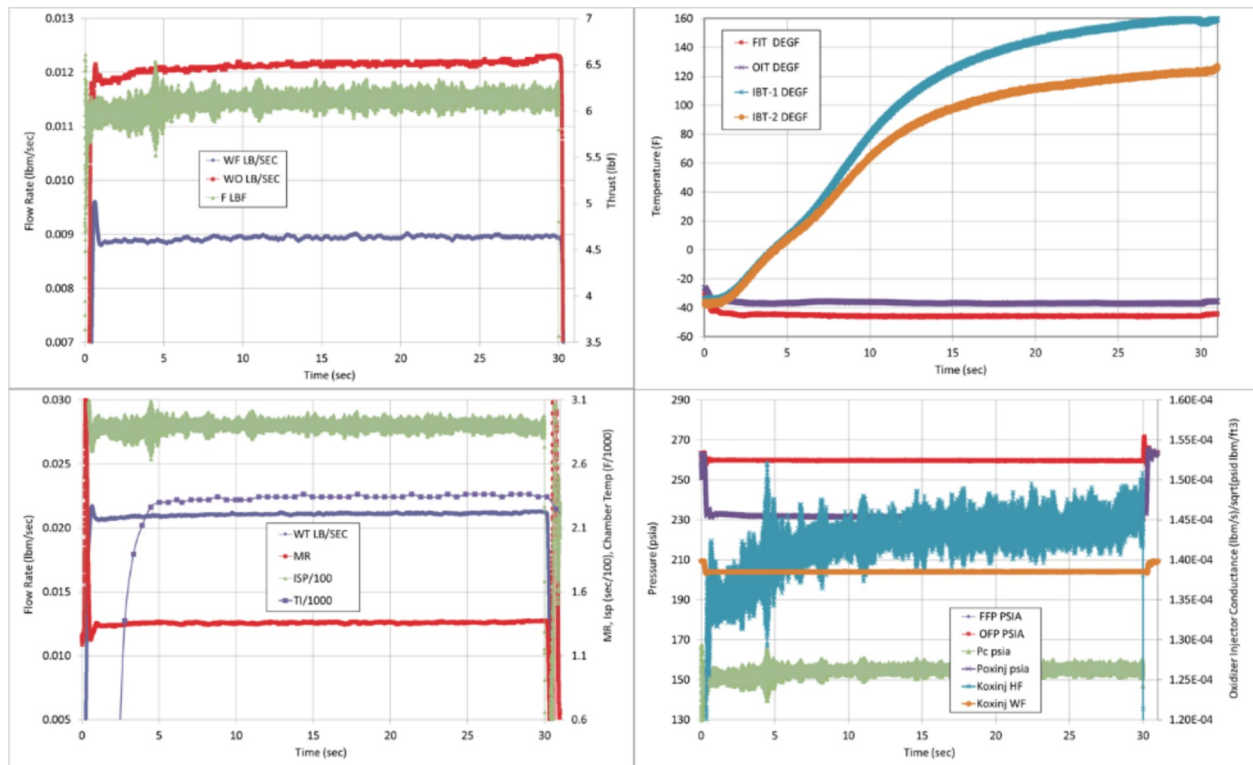


Fig. 6 Test A1-56564 (-40°C) Data Showing Steady Performance

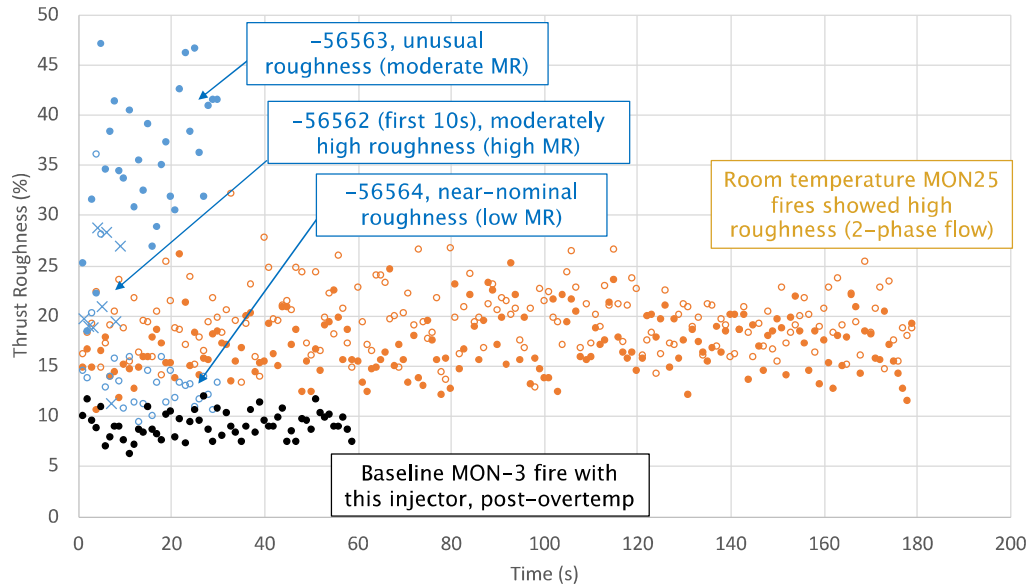


Fig. 7 Combustion Roughness with MMH/MON25 and with MMH/MON3

however. Note that Fig. 7 also shows the roughness measured with the same injector hardware during an earlier MMH/MON 3 test. The cause for this variation in roughness was not apparent. It is reminiscent, however, of similar roughness called “periodic ignitions” reported in reference [14] associated with the presence of gas, not necessarily from boiling. This possibility raises the issue of the lack of understanding of the exact chemical nature of MON25 at these conditions: how much of the NO or other intermediate species remain unreacted as gas in solution, for instance.

In order to confirm the expectation of high performance for the test hardware used, the three -40°C runs were compared against the performance obtained earlier with the same injector hardware with MMH/MON3 propellants at ambient temperature. These graphs are shown in Fig. 8. While the thrust of the -40°C MMH/MON25 DST-12 is almost perfectly on the trend line for the MMH/MON-3 firings, there is a significant specific impulse decrease of approximately 15 seconds from an extrapolation of the same injector’s performance with 20°C MMH/MON3. It could be argued that the 20% higher thrust data implies a 20% reduction in combustion residence time and therefore at least a potential for lower performance. But the MMH/MON3 EC DST engine also runs at the higher thrust level and it yields well in excess of 300 seconds Isp, consistent with the Isp trend for MON3 projected in Fig. 8. Thus, the present performance shortfall appears caused by the different oxidizer, and not by out-of-family test hardware. This conclusion applies only to the DST-series engine and is in contrast with the results of MMH/MON25 testing of the LTT [7, 8], as already discussed.

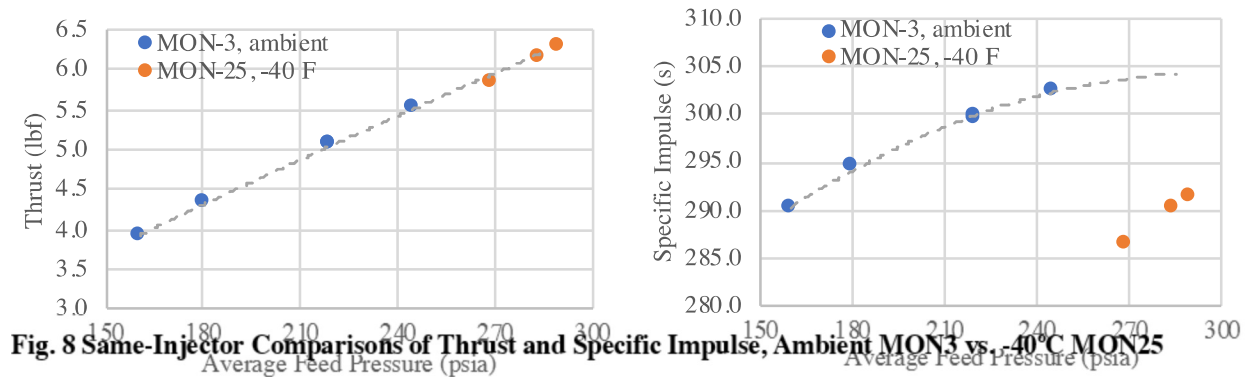


Fig. 8 Same-Injector Comparisons of Thrust and Specific Impulse, Ambient MON3 vs. -40°C MON25

VII. Conclusions

The aim of this test program was to find out whether a flight-proven MMH/MON3 engine would operate acceptably with -40°C MMH/MON25 propellants such that it could be flight-qualified for a Cold Prop application. There were three issues with the engine as tested: low Isp performance, incomplete combustion as evidenced by the facility vent emissions, and large and unexplained variations in combustion roughness at -40°C. The team held an extensive Peer Review of the data and further discussions and analyses to determine if the engine could be modified to address these issues and still retain the flight heritage of the DST-series engines to justify expedited flight qualification and subsequent mission infusion. It was ultimately concluded that, given the interdependency of the injector and the thrust chamber operation, any injector modifications tailored to resolve the observed deficiencies would likely impact other engine aspects in unpredictable ways. The risk and expense of embarking on such a redesign would be tantamount to developing a new engine and negate the flight heritage benefit sought.

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